

1N-20

198093

14P

Evaluation of Externally Heated Pulsed MPD Thruster Cathodes

Roger M. Myers
Sverdrup Technology, Inc.
Lewis Research Center Group
Brook Park, Ohio

Matthew Domonkos
The University of New Mexico
Albuquerque, New Mexico

and

Alec D. Gallimore
The University of Michigan
Ann Arbor, Michigan

Prepared for
Lewis Research Center
Under Contract NAS3-25266

NASA
National Aeronautics and
Space Administration

N94-21565

Unclas

G3/20 0198093

(NASA-CR-191090) EVALUATION OF
EXTERNALLY HEATED PULSED MPD
THRUSTER CATHODES Final Report
(Sverdrup Technology) 14 p

Evaluation of Externally Heated Pulsed MPD Thruster Cathodes

Roger M. Myers¹
Sverdrup Technology, Inc.
Lewis Research Center Group
Brook Park, Ohio 44142

Matthew Domonkos²
The University of New Mexico
Albuquerque, New Mexico 87131

Alec D. Gallimore³
The University of Michigan
Ann Arbor, Michigan 48109

Abstract

Recent interest in solar electric orbit transfer vehicles (SEOTVs) has prompted a reevaluation of pulsed magnetoplasmadynamic (MPD) thruster systems due to their ease of power scaling and reduced test facility requirements. In this work the use of externally heated cathodes was examined in order to extend the lifetime of these thrusters to the 1000 to 3000 hours required for SEOTV missions. A pulsed MPD thruster test facility was assembled, including a pulse-forming network (PFN), ignitor supply, and propellant feed system. Results of cold cathode tests used to validate the facility, PFN, and propellant feed system design are presented, as well as a preliminary evaluation of externally heated impregnated tungsten cathodes. The cold cathode thruster was operated on both argon and nitrogen propellants at peak discharge power levels up to 300 kW. The results confirmed proper operation of the pulsed thruster test facility, and indicated that large amounts of gas were evolved from the BaO-CaO-Al₂O₃ cathodes during activation. Comparison of the expected space charge limited current with the measured vacuum current when using the heated cathode indicate that either that a large temperature difference existed between the heater and the cathode or that the surface work function was higher than expected.

Nomenclature

A_{ic}	inner cathode surface area, m ²	P_h	heater power, W
A_h	heater surface area, m ²	R_a	anode radius, cm
I_{sc}	space charge limited current, A	R_{ic}	inner cathode radius, cm
J	discharge current, A	R_{oc}	outer cathode radius, cm
k	heat conduction coefficient, W/m ² C	T_{ic}	inner cathode surface temperature, °C
l	electrode lengths, cm	T_{oc}	outer cathode surface temperature, °C
\dot{m}	propellant mass flow rate, g/s	T_h	heater temperature, °C

¹Propulsion Engineer, member AIAA

²Undergraduate student, member AIAA

³Assistant Professor, member AIAA

V	voltage, V
β^2	function of R_a/R_c
ϵ_h	heater emissivity
ϵ_{ic}	inner cathode surface emissivity

Introduction

The simplicity and robustness of magnetoplasmadynamic (MPD) thrusters may make them an attractive alternative for primary propulsion systems on both near Earth and planetary solar electric spacecraft. However, while demonstrated specific impulse levels and power handling capabilities are adequate for these missions, demonstrated steady-state thruster efficiency and lifetime are still below acceptable levels for these missions.¹⁻⁴ A recent effort examining the possibility of using pulsed thrusters, with a high instantaneous power and having a duty cycle determined by the available spacecraft power, showed that the system level issues associated with pulsed thruster technology did not preclude their application, and showed that pulsed thrusters offered several benefits from a system perspective.⁵ These benefits include ease of power scaling between 10 and 40 kW, so that the same system could be used on different spacecraft having a broad range of power levels, the ability to accommodate spacecraft power level changes without changes in efficiency and specific impulse, improved testing capability in available facilities, and a reduction of facility impacts during thruster integration testing. In addition to these potential benefits, pulsed MPD thrusters have already been successfully integrated on experimental spacecraft, and two systems have been flown.⁶⁻⁹ These flights included one on the space shuttle during the Space Experiments with Particle Accelerators (SEPAC) test in 1983 and a set of two thrusters on the MS-T4 spacecraft launched by Japan in 1980. The pulse power levels for these thrusters were 2 MW and 100 kW for the shuttle and MS-T4 flights, respectively, and both used discharge durations over 1 ms. Another flight is planned on the Japanese Space Flyer Unit (SFU) which will be launched in 1994.¹⁰ The MPD thruster system for the SFU flight has passed the electromagnetic interference, thermal, and vibration qualification tests.

Achieving the potential mission benefits requires improvement of MPD thruster efficiency at specific

impulses below 5000 s and the demonstration of a long-life pulsed MPD thruster. Recent work at Princeton University,¹¹ the Los Alamos National Laboratory,¹² and the Lewis Research Center¹³ has shown that significantly improved performance can be achieved over that observed in early experiments by using hydrogen and deuterium propellants. While not yet at a level adequate for orbit-raising missions, the performance improvements were obtained without any thruster optimization. Pulsed MPD thruster lifetime has been limited to date by cathode erosion, which, because of the cold cathode nature of the discharges studied, has precluded demonstration of thruster lifetimes greater than 100 to 200 hrs. While there have been several recent proposals to externally heat a cathode to a temperature sufficient for thermionic emission,^{14,15} this approach has not yet been successfully demonstrated in a pulsed MPD thruster. However, preliminary studies by Andrenucci¹⁵ in which the cathode was heated to incandescence, though not to thermionic emission, showed significant differences from the results obtained using simple cold cathodes. In addition, Japanese system level tests in which a thruster with a low work function cathode was pulsed at rates of up to several Hertz to heat the cathode, showed that there were cases in which the thruster discharged without use of a high voltage ignitor.¹⁶ This result shows that the high voltage ignitor may be eliminated by using a heated cathode.

This paper presents the first results of an effort to evaluate externally heated cathodes for application to pulsed thruster technologies. This effort included construction of the test facility, PFN, and thruster. Following a description of the experimental facilities and diagnostics, a summary of results obtained to date is presented, including cold cathode tests and preliminary cathode heating results. Finally, a summary of conclusions is presented.

Experimental Apparatus

Test Facility and Thruster Power System

The test facility consisted of an 0.76-m diameter by 1-m tall vacuum tank pumped by a 100 l/s turbomolecular pump and two mechanical pumps. This facility has been successfully used for experiments using externally heated, impregnated tungsten hollow

cathodes.^{17,18} The facility base pressure with the turbomolecular pump was approximately 2.6×10^{-4} Pa. The tank had various viewing ports and was accessed through the top for these experiments. The thruster was bolted to the bell jar lid as shown schematically in Fig. 1.

The thruster was powered using an 10 stage, 2.5 kJ pulse forming network (PFN) which delivered a ~ 5 ms duration current pulse. The PFN energy storage capability was selected to approximate the requirements for SEOTV missions at power levels between 10 and 40 kW based on the results of a system level study.⁵ The pulse duration was chosen to permit studies of propellant utilization efficiency. The PFN consisted of twenty 350 V, 2.5 mF aluminum electrolytic capacitors with 10.8 mH inductors made of multi-strand wire. The PFN discharge was controlled using an silicon controlled rectifier (SCR) with a 1000 V standoff capability. No matching resistor was used in the circuit, and no ringing has been observed in discharges to date. While no effort was made to reduce the PFN mass or volume, the entire PFN fit within volume of 0.5 m³ inside an instrument rack. For these initial experiments, the PFN was charged manually using a 300 V, 20 A supply. A 300 ohm resistor in series with the charging supply limited the current to the PFN to less than 1 A, and resulted in a 3 minute long charging time for the PFN.

A 1 kV, 2 A D.C. supply was used to ignite the thruster. This supply was turned on to over 900 V before triggering the gas valve and SCR. A pulsed ignitor similar to those used on arcjets¹⁹ was tested, but did not have sufficient energy to trigger the PFN discharge.

The cathode heater was designed so that the heater power supply could be connected between one side of the heater and the cathode as shown in Figure 2. The 100 V, 8 A supply was isolated from ground using a 1 kV isolation transformer.

The thruster voltage was measured using x 100 voltage probes connected to a differential amplifier with unity gain. Thruster current was measured using a Rogowski coil having a response of 4 mV/A. Both voltage and current were recorded on a digital oscilloscope. An optical pyrometer was used to measure the cathode surface temperature, and thermocouples were placed on various parts of the

thrusters and facility to ensure that no damage was done to the hardware during cathode heating.

Propellant injection was controlled using an automotive fuel-injector valve. These valves are typically qualified for up to 6×10^8 , 2.5 ms duration pulses and are rated for operation at 148 °C.²⁰ While the "dead volume" between the valve and the thruster must be minimized in an operational thruster in order to maximize the propellant utilization efficiency, no effort to do this was undertaken for these experiments. The preliminary experiments reported below were performed using a fuel-injector valve with a 3×10^{-3} cm² orifice area, which severely increased the length of the propellant pulse. The valve was fed from a 313.6 cm³ plenum. The propellant flow rate was calibrated for both argon and nitrogen at plenum pressure of 60 psi by closing off the plenum feed line, firing the valve 100 times for each gas, and recording the pressure drop. The results were 0.35 g/s \pm 0.05 g/s for argon and 0.26 g/s \pm 0.04 g/s for nitrogen. The large uncertainties reflect errors in the pressure drop measurement.

The firing of the thruster was controlled using a set of adjustable delay pulse generators. As shown schematically in Figure 2, the pulse generators triggered the automobile fuel-injector valve and the PFN in a preset sequence. The capability of varying the delay between the triggers was required to permit optimization of the propellant utilization efficiencies.

MPD Thrusters

Two self-field MPD thrusters were tested for this work. The first, used primarily for cold cathode testing and facility validation, consisted of an 4 cm diameter, 5 cm long BaO-CaO-Al₂O₃ impregnated porous tungsten cathode surrounded by a 10.2 cm diameter stainless steel anode. As shown in Fig. 3, the cathode was hollow with an 1.3 cm diameter hole in the center. A molybdenum tube was brazed to one end to permit attachment of power leads to the cathode using a clamp. This cathode had been used previously for steady-state hollow cathode MPD thruster tests, so it was not known whether any of the low work function impregnate remained in the cathode. Nevertheless, the cathode was used to evaluate heater designs and facility requirements for the heated cathode experiments. A first-generation internal heater was built by threading a 1.3 cm diameter boron nitride rod with 5 threads per cm, wrapping 0.051 cm diameter tantalum wire into the

threads, and inserting it into the cathode. The heater power lead extended out the molybdenum tube base. A 2.5 cm thick boron nitride plate served as thruster backplate, propellant distributor, and cathode support. Propellant was fed into an a 3 mm wide groove machined into the boron nitride and entered the thruster through an annulus at the cathode base. The fuel-injector valve was connected to the thruster using approximately 20 cm of 0.63 cm diameter stainless steel tubing. A 1 cm long ceramic tube was connected between the stainless steel tube and the fuel-injector valve to reduce heat conduction to the valve.

The second thruster, used to evaluate heated cathodes, was built using a modified heated cathode and support structure. As shown in Figure 4, the 3.8 cm diameter, 6.35 cm length BaO-CaO-Al₂O₃ impregnated porous tungsten cathode in this thruster had a 2.54 cm diameter hole extending its length, with an additional 0.7 cm diameter hole in the cathode face. The cathode had not been used previously. The cathode heater was similar in design to that used for tests in the cold cathode thruster, except that it was 2.54 cm in diameter and had 10 threads per cm. The heater temperature was measured using a platinum - 13% rhodium/platinum thermocouple placed near the inner surface of the cathode as shown in the figure. The cathode support and propellant injection were also modified. The cathode in this thruster was supported using an annular stainless steel clamp which surrounded a flange on the cathode base. The clamp was bolted to a large stainless steel disc as shown in Figure 4. The boron nitride backplate was also bolted to this stainless steel disc. Because of the large operating temperature range and the relatively high thermal expansion of stainless steel, care was taken to leave room for expansion of the stainless steel. The propellant injection was modified to include a 0.3 cm long ceramic isolator in the feed line and to use 10 cm of 0.32 cm diameter stainless steel tubing.

Extreme care was taken during the heated cathode experiments to eliminate all possible sources of outgassing. This included elimination of all tape and epoxy, and cleaning all thruster components (except the cathode) and facility surfaces with acetone and alcohol before the final assembly. In addition, care was taken not to touch the impregnated cathode material during assembly. While the propellant feed system was flushed several times with 99.999% pure argon, it was not heated to ensure adequate bake-out of adsorbed oxygen.

Results

The cold cathode thruster was first used to validate the design of the PFN and operation of the delay timers and driving circuits. Data were taken to characterize the cold cathode thruster and to establish the behavior of the facility when the cathode was heated. Following these tests, the heated cathode thruster was installed to evaluate its performance.

Cold Cathode Experiments

It was found using the thruster shown in Figure 3 that an ignitor voltage of 800 V was required to trigger the discharge for both argon and nitrogen propellants with PFN charging voltages between 75 and 260 V. PFN voltages below these values could not be used because the ignitor discharge voltage never dropped below 75 V, precluding opening of the SCR at lower voltages. Typical thruster voltage and current traces for the cold cathode discharges using argon and nitrogen propellant are shown in Figures 5 and 6, respectively. Both traces were clearly quasi-steady, though the slow rise of the current resulted in a stabilization time of ~ 1 ms. After the peak current level was reached at ~ 1 ms, the current dropped by ~ 20% over the next 2.5 ms. The voltage trace shown for nitrogen exhibited a sudden transition after ~ 0.6 ms, dropping by ~ 40 V to the quasi-steady value of 81 V. Similar transitions were occasionally observed for argon propellant, but it was not possible to determine their cause. After the discharge the PFN was typically still charged to between 30 V and 75 V depending on the initial charging voltage. These tests showed that the PFN design was successful, though the slow rise time and lack of ringing behavior indicated that the circuit impedance was too high, yielding a slightly over-damped PFN-thruster network.

The discharge current was varied between 400 A and 3200 A by varying the PFN charging voltage. The resulting voltage - current characteristics for argon and nitrogen at 0.35 g/s and 0.26 g/s, respectively, are shown in Figure 7. The discharge voltage for argon increased from 60 V at 400 A to 110 V at 2750 A. The voltage using nitrogen propellant increased from ~ 55 V at 1000 A to ~ 85 V at 3200 A.

The operation of self-field MPD thrusters has been shown to correlate with the parameter J^2/\bar{m} for a broad range of operating conditions.²¹ These operating

conditions correspond to J^2/\dot{m} from $6 \times 10^8 \text{ A}^2\text{-s/kg}$ to $2.1 \times 10^{10} \text{ A}^2\text{-s/kg}$ for argon, and from $3 \times 10^9 \text{ A}^2\text{-s/kg}$ to $4 \times 10^{10} \text{ A}^2\text{-s/kg}$ for nitrogen. Previous work at Princeton²¹ has yielded similarly shaped voltage-current characteristics for argon propellant over this range of J^2/\dot{m} .

The cold cathode thruster was also used to assess the heater design and facility requirements for heated cathodes. Temperatures recorded using the optical pyrometer and thermocouples around the thruster mounting structure were used to estimate power losses from the cathode. The results indicated that approximately 350 W would be needed to achieve a cathode surface temperature of $\sim 1050^\circ \text{C}$, and showed that the threaded boron nitride heater design could survive the thermal cycling required for these experiments. Thermocouple measurements of the fuel-injector valve temperature showed that it stayed below its 148°C limit.

A major problem with these experiments was the small orifice size of the fuel-injector valves that were used. This issue had several consequences. First, it resulted in a very gradual increase in propellant flow rate during the pulse. Second, because extremely high plenum pressures were required to achieve the desired flow rates there was no truly quasi-steady propellant pulse. As discussed by Jones,²² the orifice area at the thruster must be less than $1/2$ the area of the valve orifice to achieve a true quasi-steady gas pulse. It was impossible to achieve this area ratio for the fuel-injector valves used in this study, showing that a first step in improving the results would be to obtain valves with much larger orifice areas.

Heated Cathode Experiments

The initial plan of this work was to follow the cathode activation procedure developed during the Solar Electric Propulsion System Program (SEPS),²³ which is supposed to yield a cathode surface work function of $\sim 2.2 \text{ V}$. This procedure is summarized in Table I. However, once the heating procedure was started it became evident it would not be possible to follow the SEPS procedure because of cathode outgassing. This problem is illustrated in Figure 8, which shows both the heater temperature and bell jar pressure as a function of time. This test was initiated immediately following an argon purge of the bell jar, which is why the initial

pressure is $1 \times 10^{-3} \text{ Pa}$. The temperatures were measured using the thermocouple in the heater shown in Figure 4. The cathode had been heated to 100°C for four hours prior to this test, which is why there was no pressure rise at that temperature. In fact, the pressure decreased over the first 2500 minutes of heating to $8 \times 10^{-4} \text{ Pa}$. However, note that when the heater temperature was increased to 400°C the pressure rose from $9 \times 10^{-4} \text{ Pa}$ to $9 \times 10^{-3} \text{ Pa}$ and took over 2 hours to recover to $3 \times 10^{-3} \text{ Pa}$. Care was taken to ensure that the bell jar pressure did not exceed $9 \times 10^{-3} \text{ Pa}$ to prevent any adverse effects on the cathode work function.²⁴ When the temperature was increased to 500°C the pressure again rose, and was allowed to recover overnight to $9 \times 10^{-4} \text{ Pa}$. The pressure rose again when the heater temperature was increased the next day, and each time the temperature was kept approximately constant until the bell jar pressure decreased. This usually took between 1 and 3 hours. While the high pressures could clearly be eliminated by using a higher pumping speed facility, the observations indicate that the larger cathode sizes used for heated cathode MPD thrusters may require different activation procedures than those used for the SEPS cathodes.

It was not possible during these experiments to achieve a discharge using the heated cathode. This was prevented by our apparent inability to achieve either a sufficiently high cathode surface temperature or a low enough work function. Direct measurement of the cathode surface temperature was precluded at the high heater temperatures because the window through which the pyrometer was pointed became contaminated. While the source of the contaminating material has not been conclusively identified, the impregnated cathode was strongly suspected due to the observed outgassing behavior. Because direct measurement was impossible, the cathode surface temperature was estimated from the measured heater power and temperature by using standard radiative heat transfer relations. The heater power required to achieve the reported heater temperatures is shown in Figure 9, where it is seen that up to 500 W was required to reach a heater temperature of 1240°C . The temperature difference between the heater and the inner cathode surface was calculated from:²⁵

$$T_{ic} = \sqrt[4]{T_h^4 - \frac{P_h}{A_h \sigma} \left[\frac{1}{\epsilon_h} + \frac{A_h}{A_{ic}} \left(\frac{1}{\epsilon_{ic}} - 1 \right) \right]} \quad (1)$$

where the emissivity of the heater and cathode were varied between 0.5 and 0.7 and the heater to cathode view factor was set to be unity because of the effectively closed surface. The outer cathode surface temperature was related to the inner cathode surface temperature by:²⁵

$$T_{oc} = T_{ic} - \frac{P_h \ln\left(\frac{R_{ic}}{R_{oc}}\right)}{2\pi Lk} \quad (2)$$

which yielded cathode outer surface temperatures between 900 °C and 1080 °C for a heater temperature of 1240 °C depending on the assumed emissivities. These results show that while the cathode could have been hot enough for adequate emission it may have been too cool, and indicate that significant effort should be expended in improving the heat transfer between the heater and the inner cathode surface.

An additional test was performed in an attempt to identify the reason for our inability to obtain a discharge. This involved measurement of the vacuum current as a function of the voltage applied between the anode and cathode and comparing the values with those expected for space charge limited current flow. At a heater temperature of 1240 °C and an anode to cathode voltage of 400 V, the current was 70 μA. The theoretical space charge limited current was calculated from:²⁶

$$I_{sc} = 1.466 \times 10^{-6} \frac{V^{3/2} L}{R_a \beta^2} \quad (3)$$

where β^2 is a function of the anode to cathode radius ratio. Ref. 26 provides a series solution for β^2 , though for this work it was approximated by:

$$\beta^2 = 0.167 \frac{R_a}{R_c} - 0.067 \quad (4)$$

which is accurate to within 10% for R_a/R_c between 1.5 and 5. For the thruster geometry tested this formulation yielded a current of 0.4 A at 400 V. The large difference between the measured and calculated currents indicated that the cathode was not emitting properly, and confirmed that either there was a large temperature difference between the heater and the cathode surface or that the work function was higher than expected.

Summary

An experimental effort was initiated to evaluate externally heated, low work function cathodes for pulsed MPD thrusters. The experimental apparatus was designed and built to approximate the sizes and discharge energies anticipated for solar electric near-Earth and planetary missions. The facility included a pulsed-forming network which delivered 5 ms duration quasi-steady current pulses and used automobile fuel-injector valves to control the propellant flow. A cold cathode thruster was successfully tested using both argon and nitrogen propellants at current levels ranging from 400 to 3250 A and peak discharge power levels of up to 300 kW. Attempts to operate a thruster using a heated BaO-CaO-Al₂O₃ impregnated porous tungsten cathode sized for the high current levels used in MPD thrusters have so far been unsuccessful. Testing to date has revealed that a large amount of gas was evolved from the cathode during activation. Heat transfer estimates show that the temperature difference between the heater and the cathode may be as high as 340 °C, showing the importance of good thermal contact between the heater and the cathode. Comparison of the theoretical space charge limited current for the tested geometry and the measured vacuum current confirm that either there was a large temperature drop between the cathode heater and the cathode surface or that the cathode work function was significantly higher than the expected value.

Acknowledgements

The authors would like to thank John Naglowsky, Fred Jent, Carl Ollick, Jerry LaPlant, Gerry Schneider, Cliff Schroeder, John Miller, and Vince Satterwhite for their invaluable assistance during this project.

References

1. Myers, R. M., Mantenieks, M.A., and LaPointe, M.R., "MPD Thruster Technology," AIAA Paper 91-3568, Sept. 1991; see also NASA TM 105242.
2. Sovey, J.S. and Mantenieks, M.A., "Performance and Lifetime Assessment of MPD Arc Thruster

Technology, Journal of Propulsion and Power, Vol. 7, No. 1, Jan. - Feb. 1991, pp. 71-83.

3. Seikel, G.R., York, T.M., and Condit, W.C., "Applied-Field Magnetoplasma Dynamic Thrusters for Orbit-Raising Missions," in Orbit-Raising and Maneuvering Propulsion: Research Status and Needs, L. Caveny, ed., Prog. in Aeronautics and Astronautics Vol. 89, American Inst. of Aeronautics and Astronautics, New York, 1984, pp. 260 - 286.

4. Buhler, R., "Plasma Propulsion for Near Earth Missions of Large Space Structures," Final Rept., Institute fur Raumfahrtssysteme, Univ. of Stuttgart, Stuttgart, Germany, May 1984.

5. Myers, R.M., Domonkos, M., and Gilland, J. H., "Low Power Pulsed MPD Thruster System Analysis and Applications," AIAA Paper 93-2391, June 1993.

6. Kuriki, K., Nakamura, K., and Morimoto, S., "MPD Thruster Test on Engineering Test Satellite," AIAA Paper 79-2071, Oct. 1979.

7. Kuriki, K., Morimoto, S., and Nakamura, K., "Flight Performance of MPD Thruster System," AIAA Paper 81-0664, April 1981.

8. Kuriki, K., "The MPD Thruster Test on the Space Shuttle," Journal of Spacecraft and Rockets, Vol. 16, No. 5, Sept. - Oct. 1979, pp. 326-332.

9. Sazaki, S., et al., "Neutralization of Beam-Emitting Spacecraft by Plasma Injection," Journal of Spacecraft and Rockets, Vol. 24, No. 3, May - June 1987, pp 227-199.

10. Shimizu, Y., Toki, K., Suzuki, H., Uematsu, K., Ostuka, T., Shiina, K., and Kunii, Y., "Development of MPD Arcjet System EM for SFU-1," IEPC Paper 91-147, Proceedings of the 22nd International Electric Propulsion Conference, Viareggio, Italy, Oct. 1991.

11. Miller, G.E. and Kelly A.J., "Hydrogen Performance Measurements," in Electric Propulsion Laboratory Progress Report MAE 1776.38, Princeton University, Dept. of Mechanical and Aerospace Engineering, July - Aug. 1992.

12. Schoenberg, K., Gerwin, R., Henins, I., Mayo, R., Scheuer, J., and Wurden, G., "Preliminary

Investigation of Power Flow and Electrode Phenomena in a Multi-Megawatt Coaxial Plasma Thruster," NASA CR 19084, March 1993.

13. Myers, R.M., "Applied-Field MPD Thruster Geometry Effects," AIAA 91-2342, June 1991; see also NASA CR 187163.

14. Myers, R. M., "MPD Thruster Technology," in Proceedings of the Second Magnetoplasma Dynamic Thruster Workshop, NASA CP 10109, May 1992, pp 36-37 and 71-90.

15. Andrenucci, M., Paganucci, F., and La Motta, G., "MPD Thruster Performance with Cathode Heating," AIAA Paper 92-3458, July 1992.

16. Yoshikawa, T., Kagaya, Y., Tahara, H., and Wasa, T., "Continuous Operation of a Quasi-Steady MPD Propulsion System with an External Magnetic Field," IEPC Paper 88-056, in Proceedings of the 20th International Electric Propulsion Conference, Garmish-Partenkirchen, Germany, Oct. 1988.

17. Soulas, G. and Myers, R.M., "Mechanisms of Anode Power Deposition in a Low Pressure Free Burning Arc," IEPC Paper 93-194, in Proceedings of the 23rd International Electric Propulsion Conference, Sept. 1993, to be published.

18. Salhi, A.H., Myers, R.M., and Turchi, P.J., "Experimental Investigation of a Hollow Cathode Discharge," IEPC Paper 93-025, in Proceedings of the 23rd International Electric Propulsion Conference, Sept. 1993, to be published.

19. Gruber, R.P., "Power Electronics for a 1-Kilowatt Arcjet Thruster," AIAA Paper 86-1507, June 1985; see also NASA TM-87340.

20. Bordewyka, T., personal communication, A.C. Rochester, Grand Rapids, MI, March 1993.

21. Gilland, J. H., Kelly, A.J., and Jahn, R.G., "MPD Thruster Scaling," AIAA Paper 87-0997, May 1987.

22. Jones, R.M., "Propellant Injection for MPD Thrusters," AIAA Paper 79-2074, Oct. 1979.

23. DePauw, J.F., "30 cm Thruster Cathode Activation," NASA TRIM No. 13, Dec. 1977.

24. Ohlinger, W., Personal Communication, Georgia Technology Research Institute, July 1993.

25. Siegel, R. and Howell, J.R., Thermal Radiation Heat Transfer, Hemisphere Publishing Co. New York, 1972.

26. Spangenberg, K.R., Vacuum Tubes, McGraw-Hill, New York, 1948.

Table I - Activation procedure used for SEPS cathodes.²³

Duration, hrs	Temperature, °C
3	500
0.5	no heat
1	1050
0.5	no heat

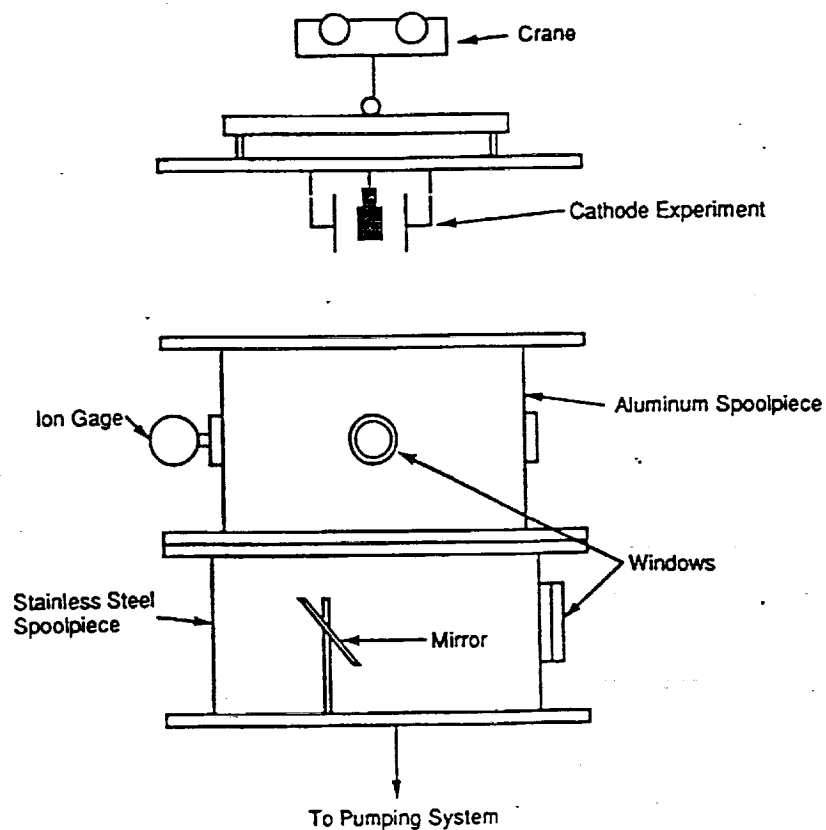


Figure 1 - Heated cathode pulsed MPD thruster test facility schematic

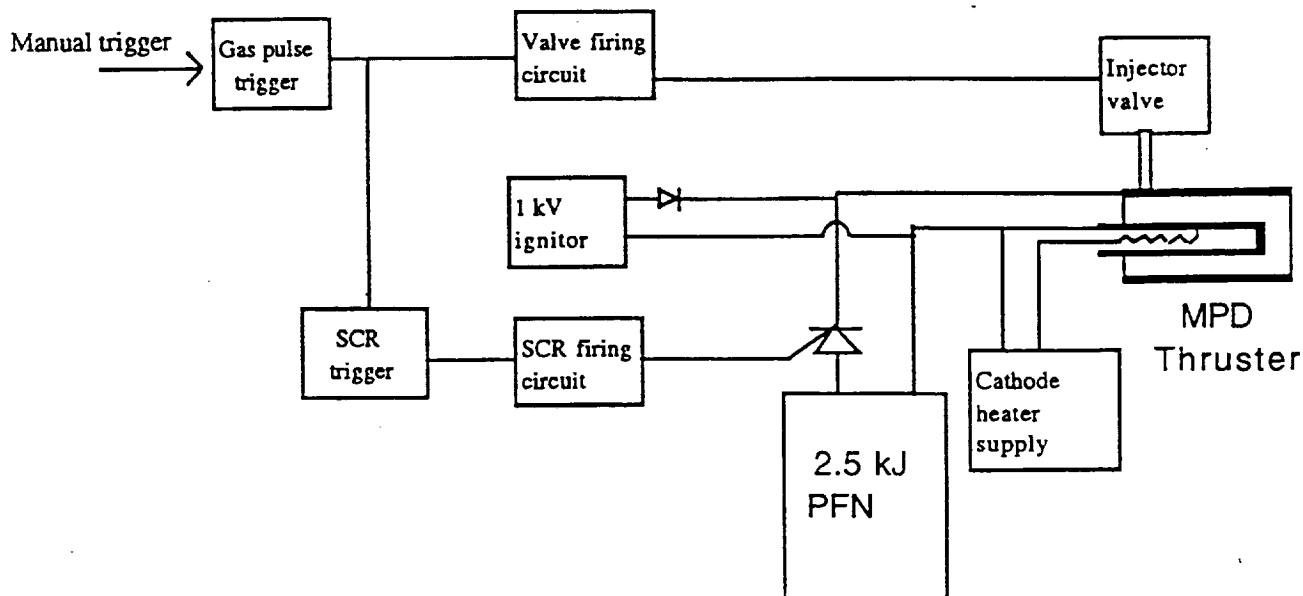


Figure 2 - Schematic of delay timers and power supplies used in pulsed MPD thruster tests.

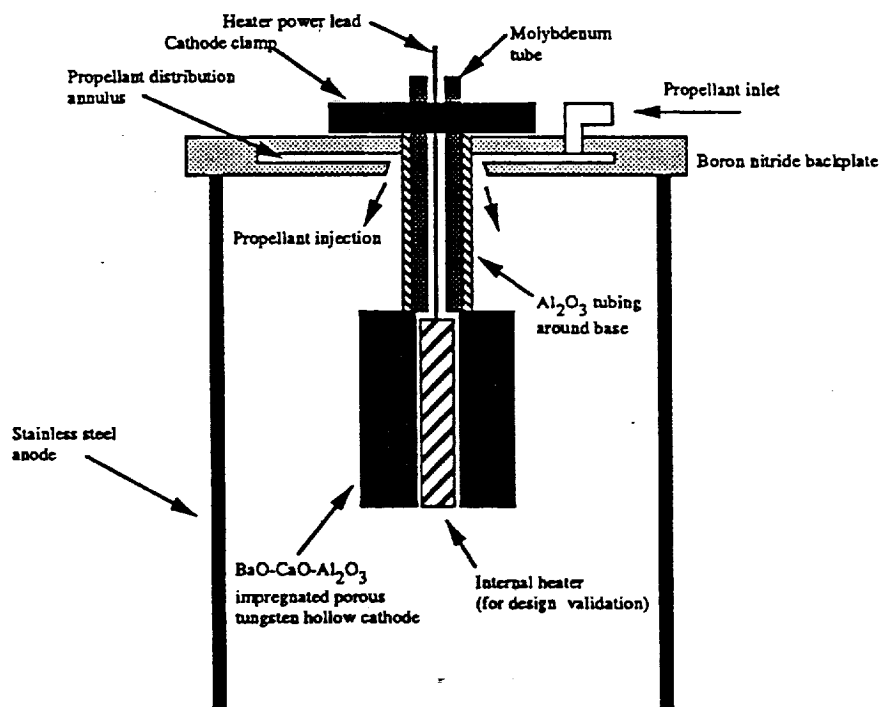


Figure 3 - Schematic of MPD thruster used for cold cathode tests and facility validation.

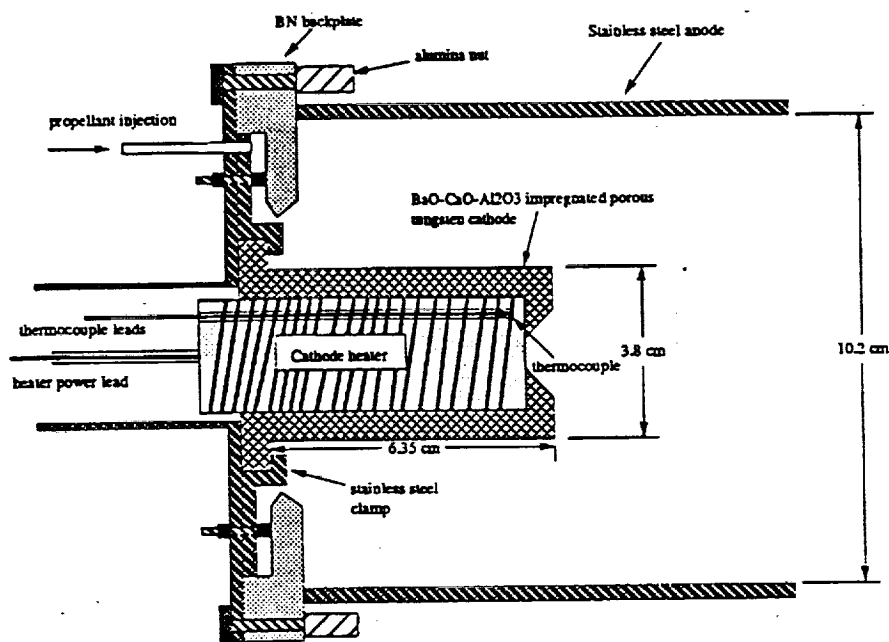


Figure 4 - Schematic of heated cathode pulsed MPD thruster.

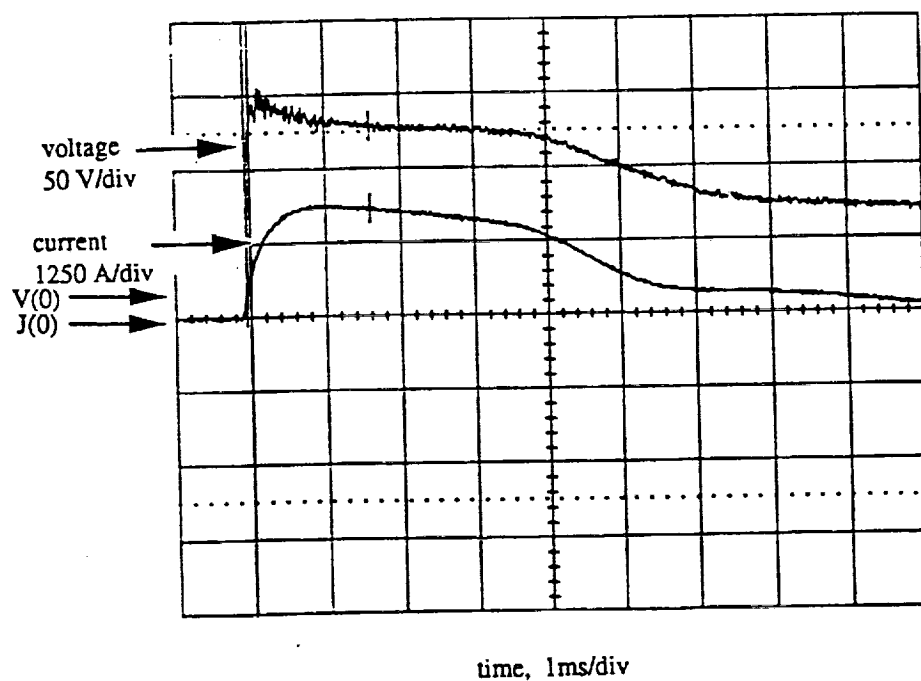


Figure 5 - Current and voltage traces for argon using the cold cathode thruster.
Current level of ~ 1840 A, voltage of ~ 109 V.

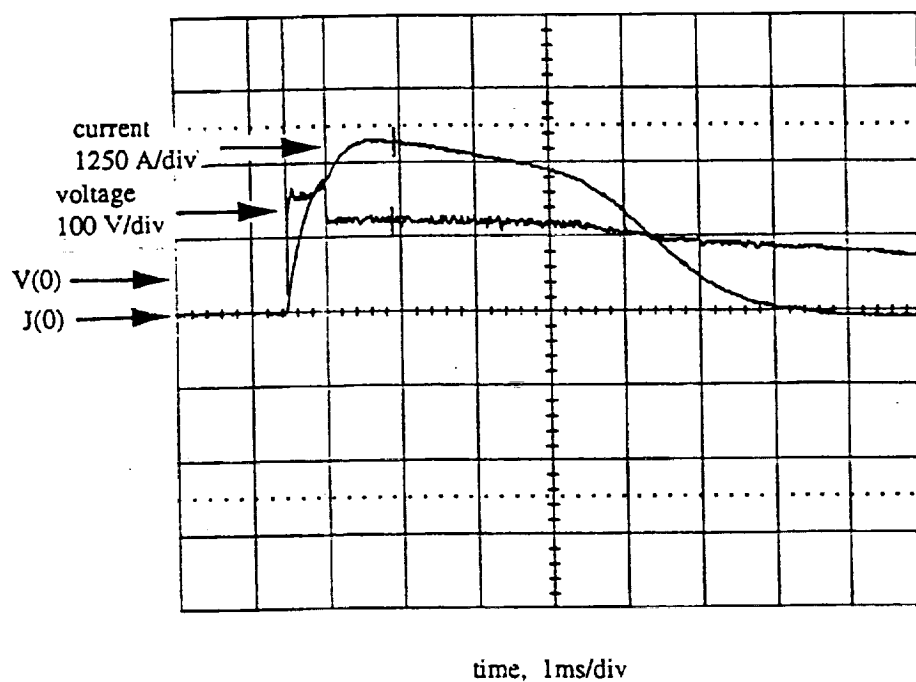


Figure 6 - Typical current and voltage traces for nitrogen propellant.
Discharge current of ~ 2850 A, voltage of ~ 80 V.

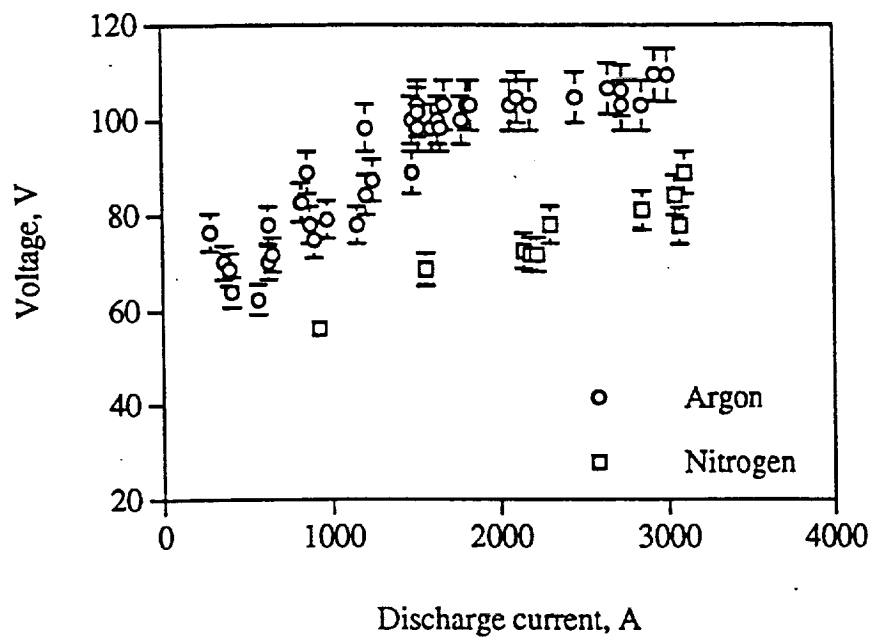


Figure 7 - Voltage - current characteristics for the cold cathode thruster operating on 0.35 g/s argon and 0.26 g/s of nitrogen.

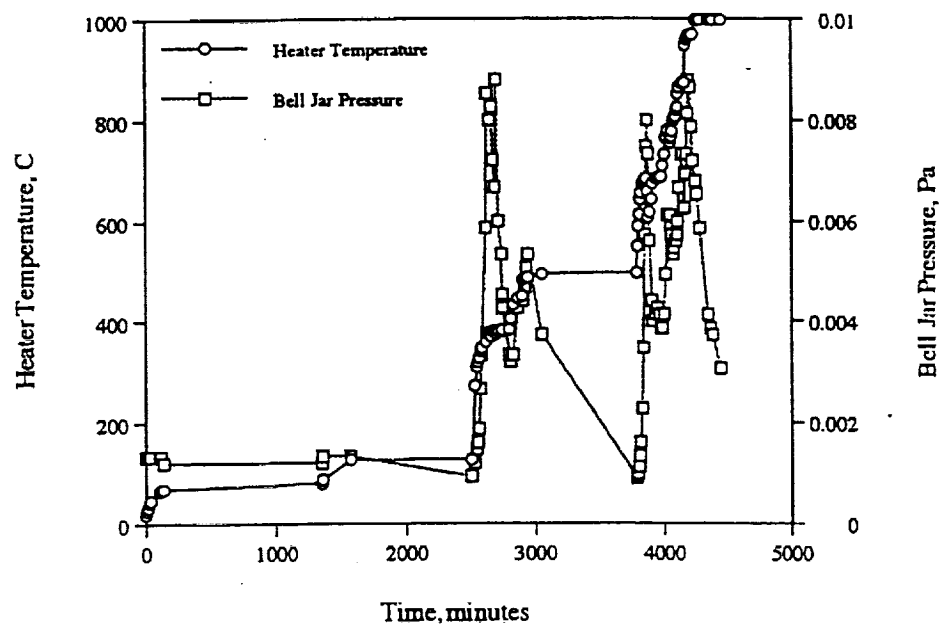


Figure 8 - Cathode heater temperature and bell jar pressure during impregnated cathode activation.

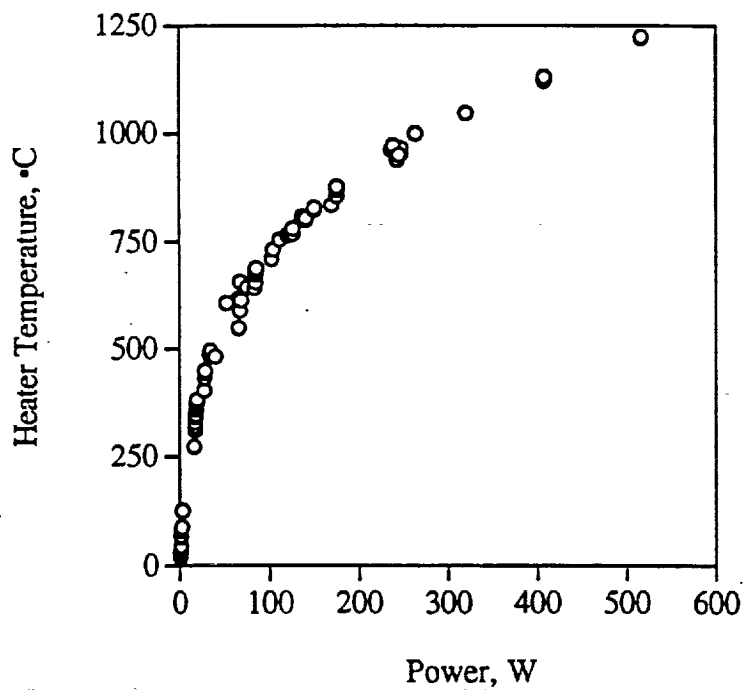


Figure 9 - Heater temperature vs. input power.

REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188	
Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.				
1. AGENCY USE ONLY (Leave blank)		2. REPORT DATE December 1993		3. REPORT TYPE AND DATES COVERED Final Contractor Report
4. TITLE AND SUBTITLE Evaluation of Externally Heated Pulsed MPD Thruster Cathodes			5. FUNDING NUMBERS WU-506-42-31 C-NAS3-25266	
6. AUTHOR(S) Roger M. Myers, Matthew Domonkos, and Alec D. Gallimore				
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) Sverdrup Technology, Inc. Lewis Research Center Group 2001 Aerospace Parkway Brook Park, Ohio 44142			8. PERFORMING ORGANIZATION REPORT NUMBER E-8293	
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191			10. SPONSORING/MONITORING AGENCY REPORT NUMBER NASA CR-191090	
11. SUPPLEMENTARY NOTES Prepared for the 23rd International Electric Propulsion Conference, sponsored by the American Institute of Aeronautics and Astronautics, Seattle, Washington, September 13-16, 1993. Roger M. Myers, Sverdrup Technology, Inc., Lewis Research Center Group; Matthew Domonkos, The University of New Mexico, Albuquerque, New Mexico 87131; and Alec D. Gallimore, The University of Michigan, Ann Arbor, Michigan 48109. Project Manager, James S. Sovey, Space Propulsion Technology Division, (216) 433-7454.				
12a. DISTRIBUTION/AVAILABILITY STATEMENT Unclassified - Unlimited Subject Categories 20 and 75			12b. DISTRIBUTION CODE	
13. ABSTRACT (Maximum 200 words) Recent interest in solar electric orbit transfer vehicles (SEOTVs) has prompted a reevaluation of pulsed magnetoplasmadynamic (MPD) thruster systems due to their ease of power scaling and reduced test facility requirements. In this work the use of externally heated cathodes was examined in order to extend the lifetime of these thrusters to the 1000 to 3000 hours required for SEOTV missions. A pulsed MPD thruster test facility was assembled, including a pulse-forming network (PFN), ignitor supply and propellant feed system. Results of cold cathode tests used to validate the facility, PFN, and propellant feed system design are presented, as well as a preliminary evaluation of externally heated impregnated tungsten cathodes. The cold cathode thruster was operated on both argon and nitrogen propellants at peak discharge power levels up to 300 kW. The results confirmed proper operation of the pulsed thruster test facility, and indicated that large amounts of gas were evolved from the BaO-CaO-Al ₂ O ₃ cathodes during activation. Comparison of the expected space charge limited current with the measured vacuum current when using the heated cathode indicate that either that a large temperature difference existed between the heater and the cathode or that the surface work function was higher than expected.				
14. SUBJECT TERMS Electric propulsion; Plasma dynamics			15. NUMBER OF PAGES 14	
			16. PRICE CODE A03	
17. SECURITY CLASSIFICATION OF REPORT Unclassified	18. SECURITY CLASSIFICATION OF THIS PAGE Unclassified	19. SECURITY CLASSIFICATION OF ABSTRACT Unclassified	20. LIMITATION OF ABSTRACT	

